BELLCOMM, INC.

SUBJECT: Saturn V Performance Capability for High Altitude Earth Orbital Missions - Case 610

DATE: October 23, 1967

FROM: P. H. Whipple

ABSTRACT

The Apollo Applications Program may eventually include synchronous and other high altitude earth orbital missions. This memorandum presents the payload capability of the Saturn V launch vehicle into circular earth orbits with altitudes in the range of 19,000 nm to 30,000 nm. The ascent mode used is a direct ascent into a Hohmann transfer ellipse, coast to apogee, and circularization at the apogee. A two-burn S-IVB is used to provide the injection into the transfer ellipse and the circularization.

The Saturn V launch vehicle can deliver large payloads into high altitude orbits. As much as 81,500 lbs can be delivered into a 19,000 nm orbit, depending upon the S-IVB propellant withheld for flight performance reserves. Similarly, up to 74,700 lbs can be orbited at an altitude of 30,000 nm. These payloads decrease approximately 1.1 lbs for each 1 lb of flight performance reserve propellant carried in the S-IVB.

A new approach in determining flight performance reserve propellant is presented. By using flight performance reserves to insure that the lowest point of the achieved orbit is above some specified minimum altitude, rather than to insure the attainment of some nominal circular orbit, the payloads for the higher altitude missions are increased. The effect of this approach is to give a relatively constant payload with increasing altitude.

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MEMORANDUM FOR FILE

I. INTRODUCTION

The Apollo Applications Program may include earth orbital missions at high altitudes. Current planning includes a mission at synchronous altitude in the early 1970's, and with program maturity, high altitude missions at other than synchronous altitude will receive consideration. This memorandum reports the results of a study of the Saturn V payload capability for circular orbits with altitudes from approximately 19,000 nautical miles to 30,000 nautical miles.

II. DISCUSSION

The ascent scheme used in this study is a direct injection into a Hohmann transfer orbit near perigee, coast to apogee, and circularization at apogee. The perigee altitude is 80 nm and the apogee altitude is chosen to be the desired circular orbit altitude. A two-burn S-IVB is assumed to provide the injection burn into the transfer orbit and the circularization burn at the transfer orbit apogee. A computer program capable of simulating this type of ascent trajectory and optimizing the payload delivered into the circular orbit was used for the trajectory simulations.[1]

A. Launch Vehicle Model

The launch vehicle data used in this study are those expected of the Saturn V in the time period of the early lunar landing missions. High altitude missions will occur considerably later than this and some improvement in launch vehicle performance can be expected. However, data in sufficient detail for trajectory simulation purposes have not been reported for launch vehicles after those scheduled for the early lunar landing missions.

1. Launch Vehicle Weight Data - Launch vehicle weights were taken from the data reported for the Saturn V SA-506 launch vehicle in Reference 2. This detailed weight statement is summarized for each stage in Appendix I. The required computer program inputs are the total launch vehicle weight at liftoff, S-II ignition, first S-IVB ignition, and S-IVB propellant depletion. These values are tabulated below.

TABLE I

LAUNCH VEHICLE WEIGHTS

	Liftoff	S-II Ignition	1st S-IVB Ignition	S-IVB Propellant Depletion
S-IC	4,949,400			
S-IC/S-II Interstage	14,170	14,170		
S-II	1,068,192	1,065,791		
S-II/S-IVB Interstage	7,645	7,645		
S-IVB	263,000	262,700	262,035	27,521
IU	4,166	4,166	4,166	4,166
LES	8,579	8,579		
	6,315,152	1,363,051	266,201	31,687

The LES weight is that expected for Mission AS-506. Other weights required as input data are given below:

Useable S-IC Propellant	4,598,260
Useable S-II Propellant	970,000
Weight Dropped Immediately After 1st S-IVB Cutoff	152
Weight Dropped Immediately Before 2nd S-IVB Ignition	733
Weight Dropped Immediately After 2nd S-IVB Cutoff	216

Weights dropped before or after S-IVB burns consist of thrust buildup or decay propellants and consumed APS propellants.

2. Launch Vehicle Propulsion Data - The launch vehicle propulsion data were based on the Saturn V AS-504 launch vehicle[3] and are given below.

	Thrust (lbs)	Weight Rate (lbs/sec)	Isp (sec)
S-IC (Sea Level)	7,664,150	28,832.5	
S-II			
(MR 4.876)	1,024,121		423.960
(MR 5.543)	1,163,649		425.143
(MR 4.690)	953,695		428.715
S-IVB	233,347		425.280

A non-propulsive S-IVB vent was assumed during the transfer orbit coast.

B. Results

1. Payloads Delivered into Circular Orbits - The results of the study are shown in Figure 1. Payload curves are shown for three values of S-IVB flight performance reserves. In each case, a 90-degree launch azimuth was used giving an orbital inclination of approximately 28.5 degrees.

For the case of zero flight performance reserves, the launch vehicle payload varies from about 81,500 lbs at an altitude of 19,000 nm to 74,700 lbs for an altitude of 30,000 nm. If the flight performance reserves are 3,000 lbs or 6,000 lbs, the corresponding payload decreases are approximately 3,300 lbs and 6,600 lbs.

Also shown in Figure 1 is the result of a trajectory simulation run made assuming a J-2S engine in the S-IVB and no flight performance reserves. The J-2S is an uprated version of the J-2, having a thrust of 265,000 lbs and a specific impulse of 431.0 seconds. An increase in payload of approximately 2,000 lbs is realized with this engine in the S-IVB.

2. Useage of the Flight Performance Reserves - Flight performance reserves (FPR) are required to offset sub-nominal launch vehicle performance. If there is no FPR allowance and the launch vehicle performance achieved during the powered flight is less than nominal, premature propellant depletion and thrust termination will occur during the circularization burn and the desired orbit will not be achieved. If adequate FPR propellant is available, the S-IVB stage can compensate for the launch vehicle sub-nominal performance by burning this additional propellant. As observed in Figure 1, the payload penalty incurred by allowing for FPR is appreciable. This penalty is approximately equal to 1.1 times the FPR weight.

Figure 2 shows the effect of an early thrust termination during the circularization burn. The variable plotted on the abscissa is the perigee altitude resulting from the early thrust termination. The apogee of the orbit is equal to the altitude of the nominal circular orbit. Plotted on the ordinate axis is the propellant shortage or additional propellant which would be required to complete the circularization burn at the time of the early thrust termination. Normally the FPR provides this additional propellant.

Consider now a desired circular orbit mission at an altitude of 30,000 nm. Assume that an analysis of the deviations in launch vehicle parameters and other pertinent quantities indicates that a FPR allowance of 6,000 lbs is necessary to insure a circular orbit of 30,000 nm in the event of sub-nominal launch vehicle performance. Also, assume that the minimum acceptable altitude due to the radiation environment and other mission constraints is 19,000 nm. Assuming no FPR allowance, the earliest expected thrust termination would result in a propellant shortage of 6,000 lbs and, from Figure 2, would give an elliptic orbit with a perigee of 20,000 nm and an apogee of 30,000 nm, an acceptable orbit with respect to the mission constraints. If the launch vehicle performance were nominal or above nominal, the desired 30,000 nm circular orbit would be achieved. If the launch vehicle performance were sub-nominal, the perigee altitude would be in the range of 20,000 nm to 30,000 nm but very likely much closer to the intended 30,000 nm. An increase in payload of about 6,600 lbs is achieved by not allowing for FPR in exchange for the acceptable risk of achieving a slightly elliptic orbit.

If the nominal circular orbit altitude were 26,000 nm, a similar situation exists. The FPR that would be required to insure a 26,000 nm circular orbit in the event of an early thrust termination, would probably be slightly less than in the previous example and is assumed to be 5,800 lbs. With no FPR allowance, the earliest expected thrust termination would result in a propellant shortage of 5,800 lbs and from Figure 2 the perigee altitude would be 17,950 nm. This is too low and sufficient FPR must be available to insure that the perigee altitude is at least 19,000 nm. From Figure 2, it is seen that a propellant shortage of 4,800 lbs results in a perigee of 19,000 nm. Therefore, a FPR allowance of 1,000 lbs is the minimum required. Decreasing the FPR from 5,800 lbs to 1,000 lbs gives an increase in payload of 5,280 lbs.

If the desired circular orbit altitude is 19,000 nm, then no reduction in FPR is permissable. To be consistent with the previous assumed values of FPR, the FPR for this case is taken as 5,450 lbs and the corresponding payload is 75,000 lbs. The values used for FPR in the previous examples are gross estimates used for illustrative purposes only.

The effect of this FPR useage philosophy is to level out the curve of payload vs. altitude as indicated in the table below. The data in this table are based on the assumed constraints used in the preceding examples.

Nominal Circular Orbit Altitude	FPR Allowance to Insure a Circular Orbit		FPR Allowance to Insure a Minimum Perigee Altitude ≤ 19,000 nm	
nm	Assumed FPR lbs	Payload lbs	FPR lbs	Payload lbs
30,000	6,000	68,100	0	74,700
26,000	5,800	69,820	1,000	75,100
22,000	5,600	72,740	3,300	75,270
19,000	5,450	75,500	5,450	75,500

As the altitude is increased from 19,000 to 30,000 nms, the payload decreases only 800 lbs compared with a payload decrease of 7,400 lbs when the conventional philosophy of FPR useage is used. In exchange for the increased payloads for the missions at the higher altitudes, the risk of slightly elliptic orbits must be accepted. However, the perigee of this orbit will probably not vary markedly from the intended circular orbit altitude and only then if the launch vehicle performance is sub-nominal, rather than nominal or above nominal.

C. <u>Launch Vehicle Modifications</u>

The modifications to the S-IVB stage to accomplish this mission are (1) the relocation of the LH_2 baffle and (2) the addition of a battery and cable. The relocation of the LH_2 baffle is necessary because the propellant level at S-IVB restart is different from the propellant level at restart in a LOR mission. These modifications are estimated by MSFC to result in a weight increase of about 100 lbs in the S-IVB stage.[4] The higher altitude missions may dictate some additional modifications to increase the IU lifetime. The current estimate of IU lifetime is about 7.5 hours. For nominal circular orbits above 26,000 nm, the mission time at S-IVB ignition for the circularization burn is greater than 7.5 hours. If the IU lifetime extension modification is necessary, the associated weight penalty is estimated to be relatively small compared with the total payload weights.

III. CONCLUSIONS

The Saturn V launch vehicle is capable of placing large payloads in high altitude orbits with only minor modifications. The launch vehicle payloads vary from 81,500 lbs into a 19,000 nm circular orbit to 74,700 lbs into a 30,000 nm altitude, assuming no flight performance reserves. Making an allowance for flight propellant reserves decreases these payloads by approximately 1.1 lbs for every 1 lb of flight performance reserves.

By modifying the philosophy of useage of flight performance reserves, additional payload can be realized. If sufficient flight performance reserves are required to insure that the minimum altitude of the final orbit is greater than some specified value, only a portion of the presently estimated flight performance reserves need be carried, with a resulting increase in payload. The final orbit for this case may be elliptic in the event of sub-nominal launch vehicle performance, but for the range of altitudes considered here, such an orbit may still be acceptable for the mission. This approach benefits the higher altitude missions more than the lower altitude missions and tends to level out the payload vs. altitude curve.

On the basis of one trajectory simulation run using the J-2S engine in the S-IVB stage only, a payload improvement of approximately 2,000 lbs can be expected for high altitude missions.

1021-PHW-dcs P. H. Whipple

Appendix I

The launch vehicle weights used in the trajectory simulations were taken from data reported for the Saturn V SA-506 launch vehicle in Reference 2.

The weight breakdown of the S-IC stage is summarized below.

Dry S-IC Stage	287,824
Residual Propellant	53,686
Separation Weight	341,510
Thrust Decay Propellant	8,948
S-IC Cutoff Weight	350,458
Available Propellant	4,598,260
Service Items Expended During S-IC Burn	682
S-IC Liftoff Weight	4,949,400

This S-IC liftoff weight is after thrust buildup and the thrust buildup propellants are not included in the available propellant or the S-IC liftoff weight. For the trajectory simulations, the service items and thrust decay propellants are assumed to be jettisoned at S-IC cutoff and the S-IC separation weight was changed to account for this.

The S-II stage weight breakdown is as follows:

Dry S-II Stage	82,524
Residuals	12,907
Separation Weight	95,431
Thrust Decay Propellant	360
S-II Cutoff Weight	95,791
Available Propellant	970,000
Thrust Buildup Propellant	1,831
S-II Ignition Weight	1,067,622
Losses Incurred During the S-IC Burn	570
S-II Weight at Liftoff	1,068,192

The losses incurred during the S-IC burn are frost and purge gas. For simulation purposes, these and the thrust buildup propellants are assumed to be jettisoned with the spent S-IC stage. The thrust decay propellant is assumed to be jettisoned with the spent S-II stage.

The S-IVB stage weight breakdown given in Reference 2 for an LOR mission is as follows:

Dry S-IVB Weight at Separation	24,579
Residuals	2,842
Separation Weight	27,421
Thrust Decay Propellant	174
2nd Burn Cutoff Weight	27,595
2nd Burn Propellant	159,171
Thrust Buildup Propellant	364
2nd Burn Ignition Weight	187,130
EPO Losses	3,869
Thrust Decay Propellant	94
1st Burn Cutoff Weight	191,093
1st Burn Propellant	70,842
Thrust Buildup Propellant and Ullage Losses	529
1st Burn Ignition Weight	262,464
Losses Prior to 1st Ignition	436
S-IVB Weight at Liftoff	262,900

Included in this weight are 236,843 lbs of cryogenic propellants. This includes 229,953 lbs for the main propulsion burns and 3,460 lbs for boiloff in earth parking orbit. For use in the computer program, these quantities were combined.

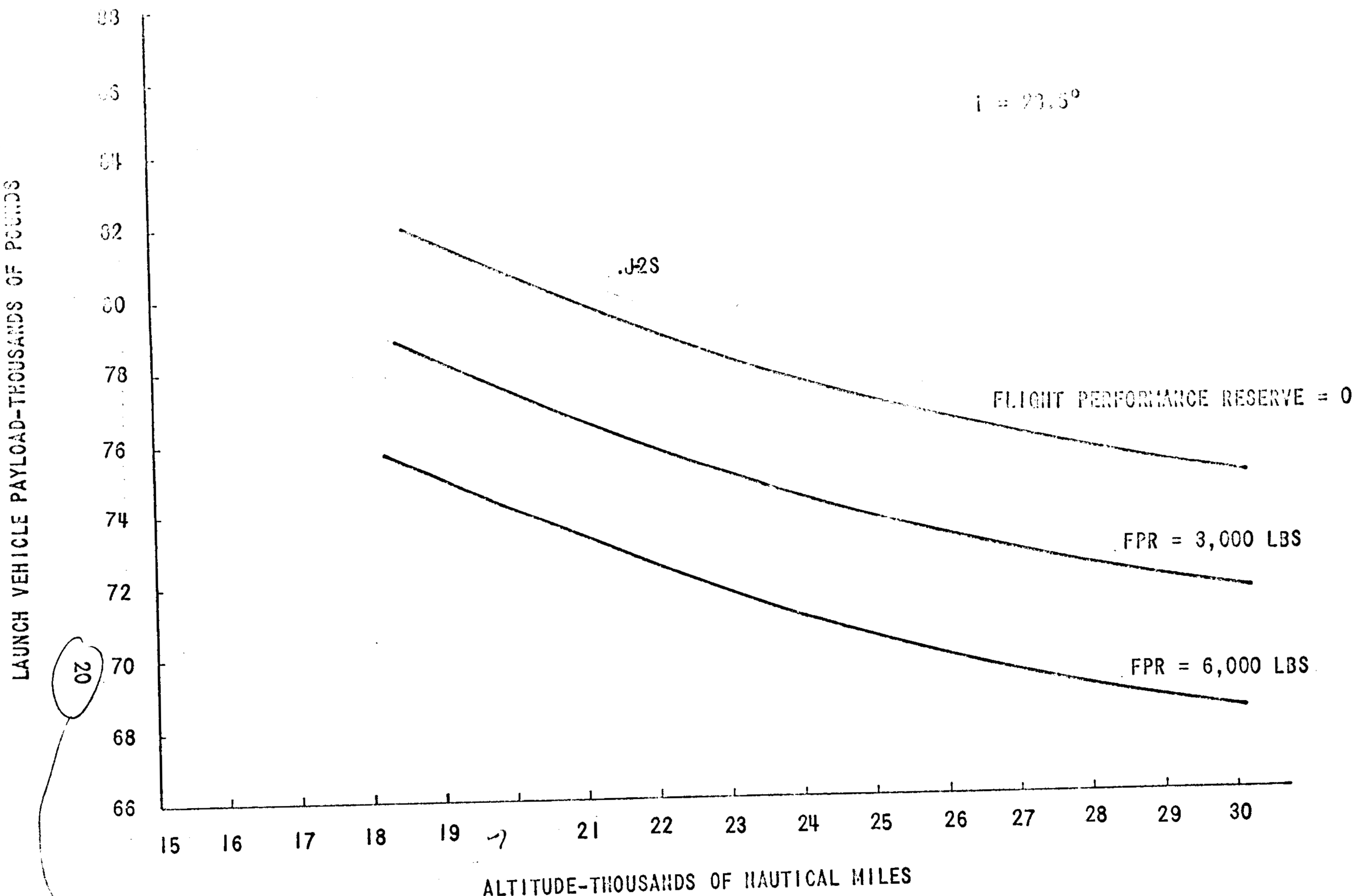

FIGURE I - SATURN V PERFORMANCE CAPABILITY

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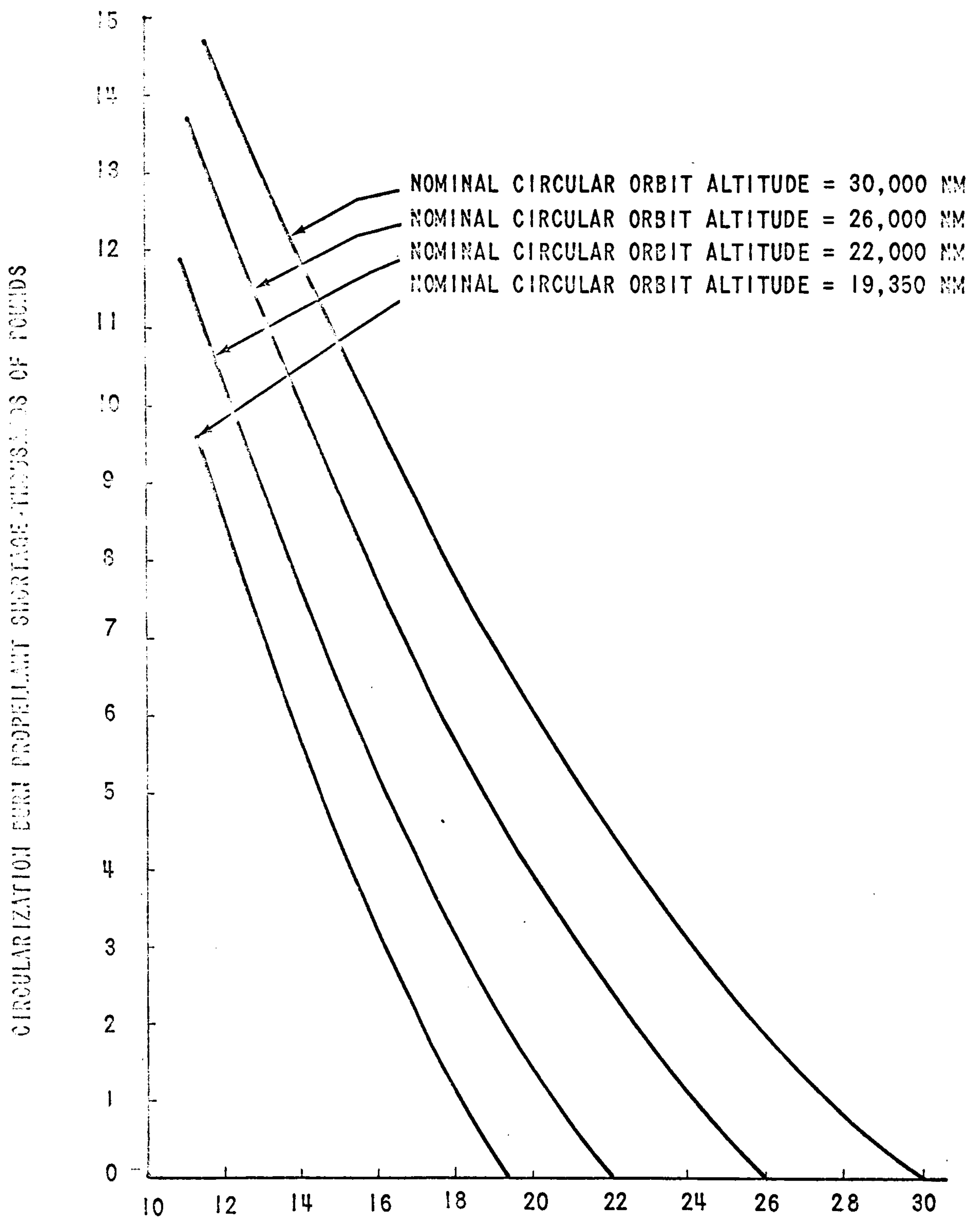

FIGURE 2 - EFFECTS OF EARLY THRUST TERMINATION DURING THE CIRCULARIAZATION BURN

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1. Whipple, P. H., Simulation of Saturn V Launch to Earth Orbit with a Hohmann Transfer and S-IVB Circularization, Memorandum for File, September 1967.

2. MSFC, Detailed Monthly Weight Status Report for the Saturn V SA-506 Launch Vehicle, R-P&VE-VAW-67-103, July 19, 1967.

3. Boeing, Saturn V AS-504 Launch Vehicle Performance Analysis, D4-15519-4, July 14, 1967.

4. MSFC, Saturn V Synchronous Orbit Study Presentation, September 16, 1966.

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